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RESEARCH MEMORANDUM

AERODYNAMIC CHARACTERISTICS OF TWO DELTA WINGS AT MACH NUMBER 4.04 AND CORRELATIONS OF LIFT AND MINIMUM-DRAG DATA FOR DELTA WINGS AT MACH NUMBERS FROM 1.62 TO 6.9

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RESEARCH MEMORANDUM

AERODYNAMIC CHARACTERISTICS OF TWO DELTA WINGS AT MACH NUMBER 4.04 AND CORRELATIONS OF LIFT AND MINIMUM-DRAG DATA FOR DELTA WINGS AT MACH NUMBERS FROM 1.62 TO 6.9

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SUMMARY

Tests were made to determine the aerodynamic characteristics at Mach number 4.04 of two delta wings of aspect ratio 2.31 with 8-percent-thick double-wedge airfoil sections having their maximum thicknesses at 18 and 60 percent chord. These tests were conducted in the Langley 9- by 9-inch Mach number 4 blowdown jet at a Reynolds number of 6.0×10^6, based on the model mean aerodynamic chord. The results were analyzed together with results obtained at lower Mach numbers in the Langley 9-inch supersonic tunnel and at Mach number 6.9 in the Langley 11-inch hypersonic tunnel for the same and similar wings. The analysis indicated that the conclusion in NACA RM L9D07, which stated that, on the basis of the data at lower Mach numbers, for wings of the same family the ratio of the experimental lift-curve slope to the theoretical two-dimensional lift-curve slope was relatively independent of Mach number for any given value of the ratio $\tan \epsilon / \tan m$ (where ϵ is the wing semiapex angle and m is the free-stream Mach angle), was also valid at Mach number 4.04. It was also found that, for double-wedge delta wings having the same maximum-thickness location, the product of the experimentally derived pressure drag and $\frac{\sqrt{M^2 - 1}}{(t/c)^2}$ (where M is the free-stream Mach number and t/c is the airfoil thickness ratio) was relatively independent of Mach number for a given value of $\tan \epsilon / \tan m$ throughout the Mach number range from 1.62 to 6.9. Thus it is shown that these methods of correlating experimental lift and pressure-drag data of delta wings provide a means of predicting wing performance at high supersonic Mach numbers from experimental results obtained at low supersonic Mach numbers.

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INTRODUCTION

The bulk of available design data for supersonic aircraft and missiles lies in the speed range up to a Mach number of roughly 2. A considerable increase has recently occurred in contemplated design speeds, and thus the need for research data in the speed range from Mach number 2 to about 5 has greatly increased. In order to provide some of the needed data, a wing program has been started in the Langley 9- by 9-inch Mach number 4 blowdown jet. This program has two objectives: first, to establish the performance at Mach number 4.04 of a number of related wings of particular interest in this speed range and, second, to develop, if possible, means of correlation with the available results for other supersonic Mach numbers. In this program the effects of plan form, thickness ratio, maximum-thickness location, and leading- and trailing-edge profiles on wing force characteristics are being investigated.

This report presents the results obtained for two 8-percent-thick wings of this program which differed only in the position of maximum thickness. One of these wings was identical to a wing previously investigated at Mach numbers of 1.62, 1.92, and 2.40 in the Langley 9-inch supersonic tunnel (ref. 1). Also presented for comparison are recently obtained preliminary results for these and similar wings from the Langley 9-inch supersonic tunnel and from the Langley 11-inch hypersonic tunnel (Mach number 6.9). These new data, together with the published results in references 1 and 2, are analyzed and discussed briefly with a view toward establishing correlation criteria.

SYMBOLS

C_L lift coefficient, $\dfrac{\text{Lift}}{qS}$

C_m pitching-moment coefficient, $\dfrac{\text{Pitching moment about wing center of area}}{qSc_r}$

C_D drag coefficient, $\dfrac{\text{Drag}}{qS}$

C_b wing-root bending-moment coefficient, $\dfrac{\text{Bending moment about wing root due to lift}}{qS\frac{b}{2}}$

α angle of attack

q	free-stream dynamic pressure
S	semispan wing area
b	wing span
c_r	wing-root chord
$\bar{c}$	wing mean aerodynamic chord, $\frac{2c_r}{3}$
$\frac{L}{D}$	ratio of wing lift to wing drag
C_{L_α}	lift-curve slope at zero angle of attack
ϵ	wing semiapex angle
m	free-stream Mach angle
M	free-stream Mach number
R	Reynolds number based on $\bar{c}$

$\beta = \sqrt{M^2 - 1}$

$\frac{t}{c}$	airfoil thickness ratio
$C_{D_{min}}$	drag coefficient at zero lift
C_{D_p}	pressure-drag coefficient, $C_{D_{min}}$ - Theoretical skin-friction-drag coefficient

APPARATUS AND TESTS

The tests were conducted in the Langley 9- by 9-inch Mach number 4 blowdown jet; this facility is described and its test-section flow calibration is presented in reference 3. The settling-chamber pressure, which was held constant by a pressure-regulating valve, and the corresponding air temperature were continuously recorded on film during each run. An external side-wall-mounted strain-gage balance was used to measure the normal force, chord force, pitching moment, and wing-root bending moment of the models. The models were mounted through a boundary-layer bypass plate (see fig. 1) which was far enough out in the stream to bypass the

tunnel-wall boundary layer. The dimensions of the semispan models are given in figure 2. The models were made of steel and had sharp leading and trailing edges and ridge lines.

Balance deflections under load necessitated about 0.10-inch clearance all around the models at the root chord. Force tests of a rectangular wing equipped with pressure orifices on its surface just outboard and inboard of the gap at the wing root showed that air flow in and out of this 0.10-inch gap caused large changes in the wing-surface pressures at angles of attack, which caused erroneous force and moment readings. A sliding-plate gap-sealing mechanism was therefore developed (fig. 3) which allowed the wing to move freely under load and reduced the effects of gap leakage to a negligible amount. Force tests were made which showed that friction between the sliding and the stationary plates did not produce any measurable forces or moments.

The Reynolds number for the tests was 6.0×10^6, based on the model mean aerodynamic chord. Because of adverse effects from choking behind the bypass plate at high angles of attack, the angle-of-attack range was limited to $\pm 14^\circ$. The tests were run at humidities below 5.0×10^{-6} pounds of water vapor per pound of dry air, which is believed to be low enough to eliminate condensation effects.

PRECISION OF DATA

The uncertainties involved in measuring the forces and moments and computing the aerodynamic coefficients and the center-of-pressure locations have been evaluated. The probable uncertainties in the data are listed below. The center-of-pressure uncertainties refer to the centers of pressure obtained by the slope method.

α, deg	± 0.05
C_L	± 0.005
C_D	± 0.001
C_m	± 0.001
C_b	± 0.0015
Chordwise center of pressure, percent $\bar{c}$	± 1
Spanwise center of pressure, percent $b/2$	± 2

RESULTS

The results are presented herein with minimum analysis and discussion in order to expedite their publication. Figure 4 presents the

variations of lift, drag, pitching-moment, and wing-root bending-moment coefficients, lift-drag ratios, and centers of pressure with angle of attack for both models. Center-of-pressure locations at zero angle of attack obtained by the slope method are indicated by short horizontal lines on the $\alpha = 0^{\circ}$ axes of figures 4(d) and 4(e). Figures 5 to 9 give the experimental and theoretical variation of several aerodynamic characteristics of wing I as functions of Mach number and of the ratio of the tangent of the semiapex angle of the wing to the tangent of the Mach angle of the free-stream flow. The values of these parameters for wing II at Mach number 4.04 are also plotted in these figures. Figures 10 and 11 present the lift-curve slopes and the minimum pressure-drag coefficients of the two wings tested at Mach number 4.04, together with data from the Langley 9-inch supersonic tunnel (ref. 1 and preliminary data at Mach numbers 1.62, 1.93, and 2.40) and from the Langley 11-inch hypersonic tunnel (ref. 2 and preliminary data at Mach number 6.9) on the same and similar delta wings.

The minimum pressure-drag coefficients at Mach numbers 1.62 to 4.04 used in figure 11 were obtained by subtracting from the experimental minimum drag coefficients values of skin-friction-drag coefficient based on theoretical laminar and turbulent skin-friction coefficients of references 4 and 5, combined by the method of reference 6, with the assumption that the boundary layer became turbulent just behind the ridge line on each wing. The minimum pressure-drag coefficients at Mach number 6.9 were obtained by subtracting from the experimental minimum drag coefficients theoretical values of skin-friction-drag coefficients computed according to the method of reference 7, with the wing considered as a flat plate and the Reynolds number based on the average square root chord of the wing.

Theoretical lift coefficients used in figures 5, 7, and 10 were obtained by the method of reference 8. Theoretical drag coefficients were made up of linear-theory pressure-drag coefficients obtained by the method of reference 9 and theoretical skin-friction-drag coefficients obtained by the method of reference 6, for the assumptions given in the previous paragraph. The theoretical drag coefficients of figure 6 at Mach numbers up to 3 were based on the average Reynolds number for wing I for the three test Mach numbers below 3. The theoretical drag coefficients at Mach numbers near 4 and 7 were based on the test Reynolds numbers in each case. Theoretical values of the maximum lift-drag ratio and the lift coefficient for maximum lift-drag ratio were obtained from reference 6.

DISCUSSION

In figure 5 it is seen that the linear theory considerably overestimated the lift-curve slope of wing I at the lower Mach numbers but gave better agreement at Mach numbers near 4.04 and 6.9. In reference 1 it was shown that, when the lift-curve slopes for wing I and for the other wings of the same family at Mach numbers of 1.62, 1.93, and 2.40 were plotted as the ratio of the measured lift-curve slope to the theoretical two-dimensional lift-curve slope against $\tan \epsilon/\tan m$ (fig. 10), the ratio was relatively independent of Mach number for a given value of $\tan \epsilon/\tan m$. The point obtained for wing I at a Mach number of 4.04 falls almost exactly on an extension of the curve through the lower Mach number data of reference 1; thus it is indicated that the first conclusion of reference 1 applies at Mach numbers as high as 4.04. The preliminary data from the Langley 9-inch supersonic tunnel on 8-percent-thick delta wings with their maximum thicknesses at 50 percent chord and the data obtained at Mach number 4.04 on the delta wing with its maximum thickness at 60 percent chord also fall on a single curve, so that this correlation provides another example of the validity of extending the Mach number range of the first conclusion of reference 1 to Mach number 4.04. The difference in maximum-thickness location from 50 to 60 percent chord probably has only a very small effect on the lift-curve slope at Mach number 4.04, since, experimentally (see fig. 5), the lift-curve slope of the wing with maximum thickness at 60 percent chord is only 5 percent greater than the lift-curve slope of the wing with maximum thickness at 18 percent chord. Moving the maximum thickness toward the rear did, however, have a large effect on the minimum drag coefficient, decreasing it by 50 percent (fig. 6). Lift data are available from reference 2 at Mach number 6.9 for a 5-percent-thick delta wing of aspect ratio 2.31 with double-wedge airfoil section having its maximum thickness at 50 percent chord. The flow over this wing is largely two-dimensional and the value of the lift-curve slope is almost equal to the linear-theory two-dimensional value $\left(\text{see fig. 10 at } \frac{\tan \epsilon}{\tan m} = 3.9\right)$.

Figure 6 shows that the linear-theory prediction of minimum drag coefficient, with its unrealistic drag peaks at flow conditions where the free-stream Mach line is coincident with the wing leading edge or coincident with the ridge line, is very inaccurate. However, when the pressure drags of the wings are obtained by subtracting an estimated skin-friction-drag coefficient and the quantity $\frac{C_{D_p}\beta}{(t/c)^2}$ is plotted against $\tan \epsilon/\tan m$ (fig. 11), correlations of the data are obtained which are relatively independent of the test Mach number as predicted by linear theory. The correlation of all points depends to some extent on the assumptions made concerning the location of boundary-layer transition and the values of the laminar and turbulent skin-friction-drag coefficients.

CONCLUSIONS

An investigation of the aerodynamic characteristics of 8-percent-thick double-wedge delta wings with 60° leading-edge sweep and with maximum thicknesses at 18 and 60 percent chord has been made at a Mach number of 4.04 and Reynolds number of 6.0×10^6. An analysis of the results of these tests, together with those of other investigations at higher and lower Mach numbers of the same and similar double-wedge delta wings, indicated the following conclusions:

1. The conclusion of NACA RM L9D07, which stated that for wings of the same family the ratio of the experimental lift-curve slope to the theoretical two-dimensional lift-curve slope was relatively independent of Mach number for any given value of the ratio $\tan \epsilon / \tan m$ (where ϵ is the wing semiapex angle and m is the free-stream Mach angle), is valid at Mach numbers up to at least 4.04.

2. For double-wedge delta wings with the same maximum-thickness location, the product of the experimentally derived pressure drag and $\frac{\sqrt{M^2 - 1}}{(t/c)^2}$ (where M is the free-stream Mach number and t/c is the airfoil thickness ratio) was relatively independent of Mach number for a given value of $\tan \epsilon / \tan m$ throughout the Mach number range from 1.62 to 6.9. This correlation provides a means of predicting pressure drag at high Mach numbers from experimental data obtained at low Mach numbers.

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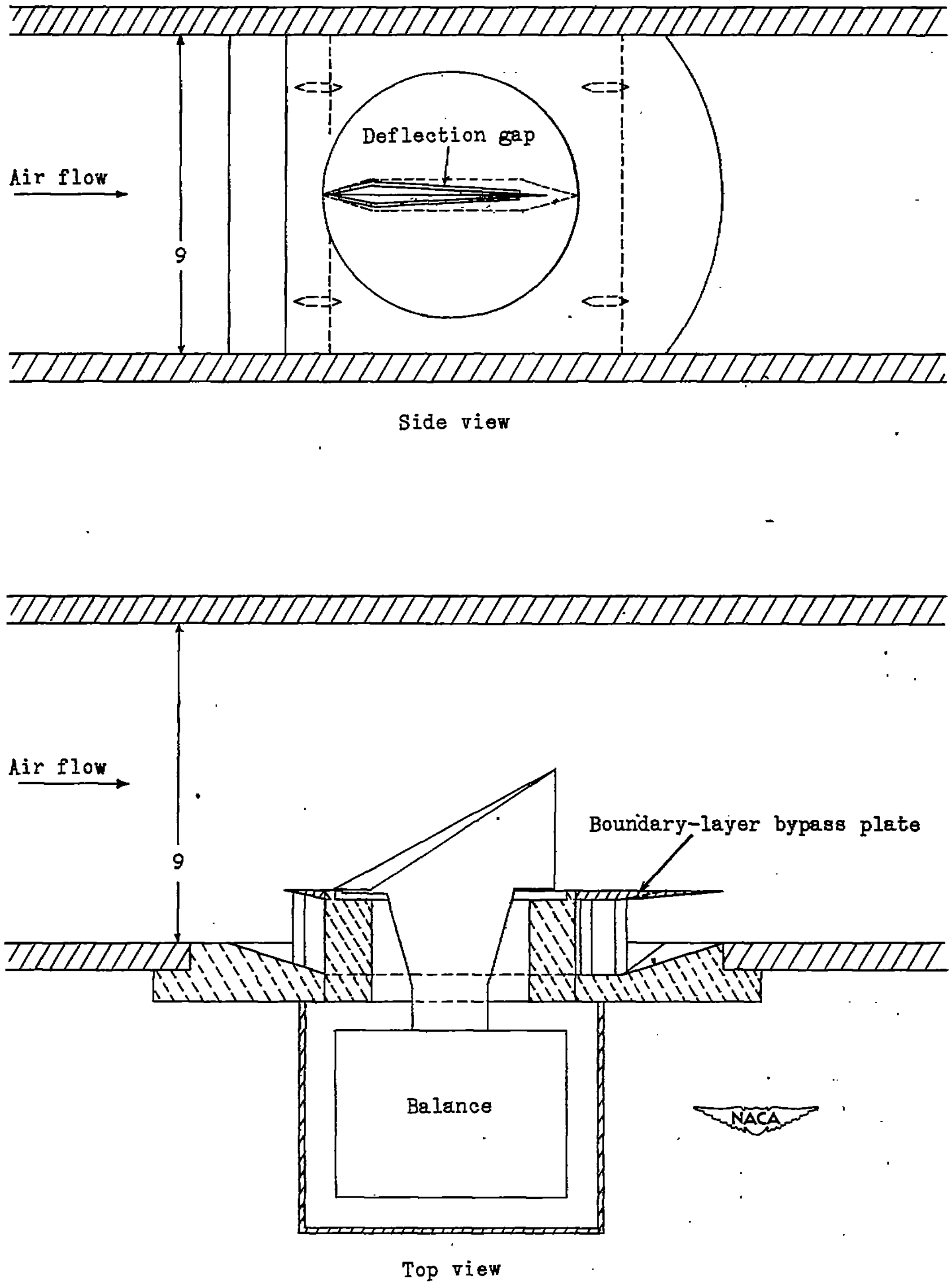

Figure 1.- Schematic diagram of test section of Langley 9- by 9-inch Mach number 4 blowdown jet and balance arrangement. All dimensions are in inches.

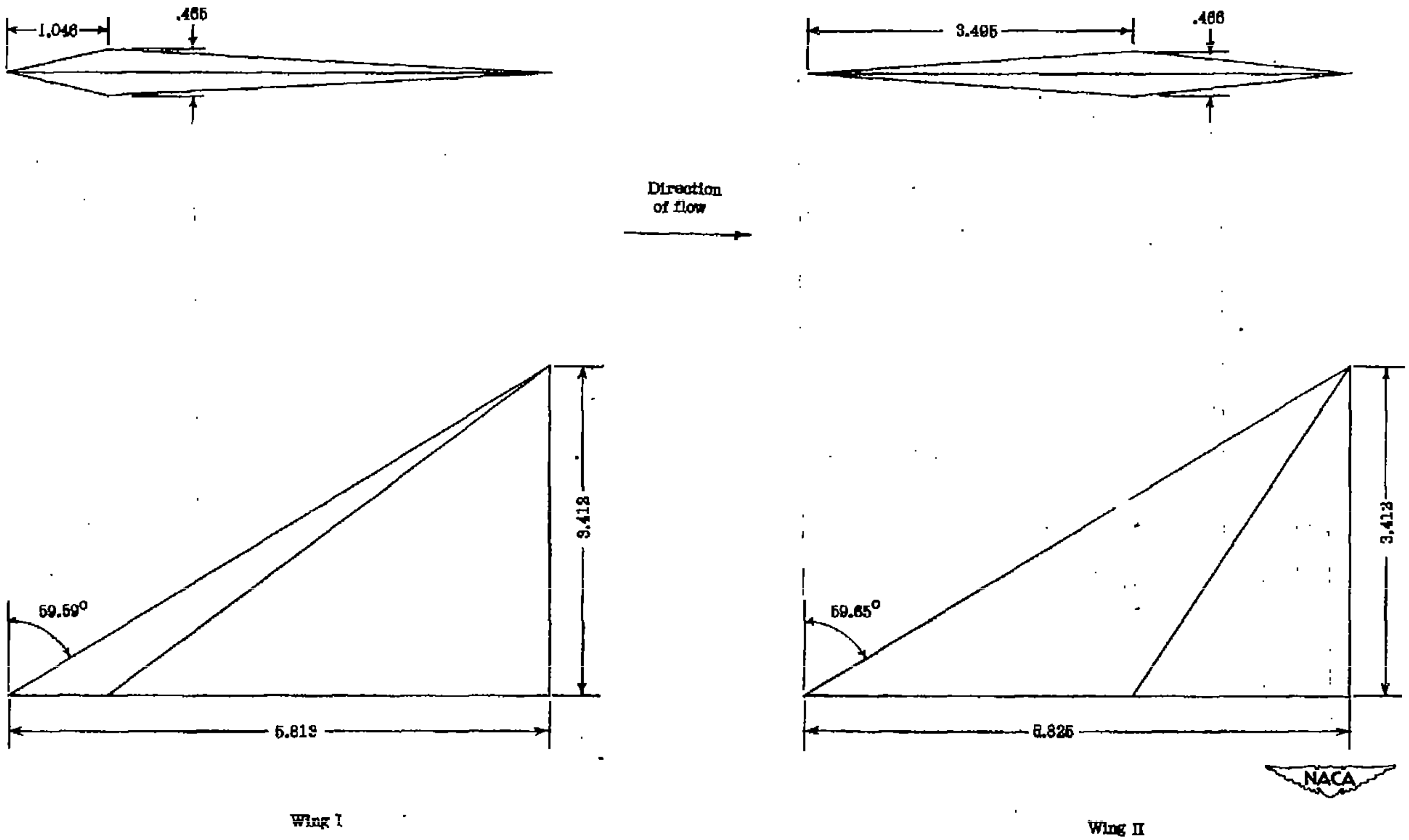

Figure 2.- Diagram of the models. All dimensions are in inches.

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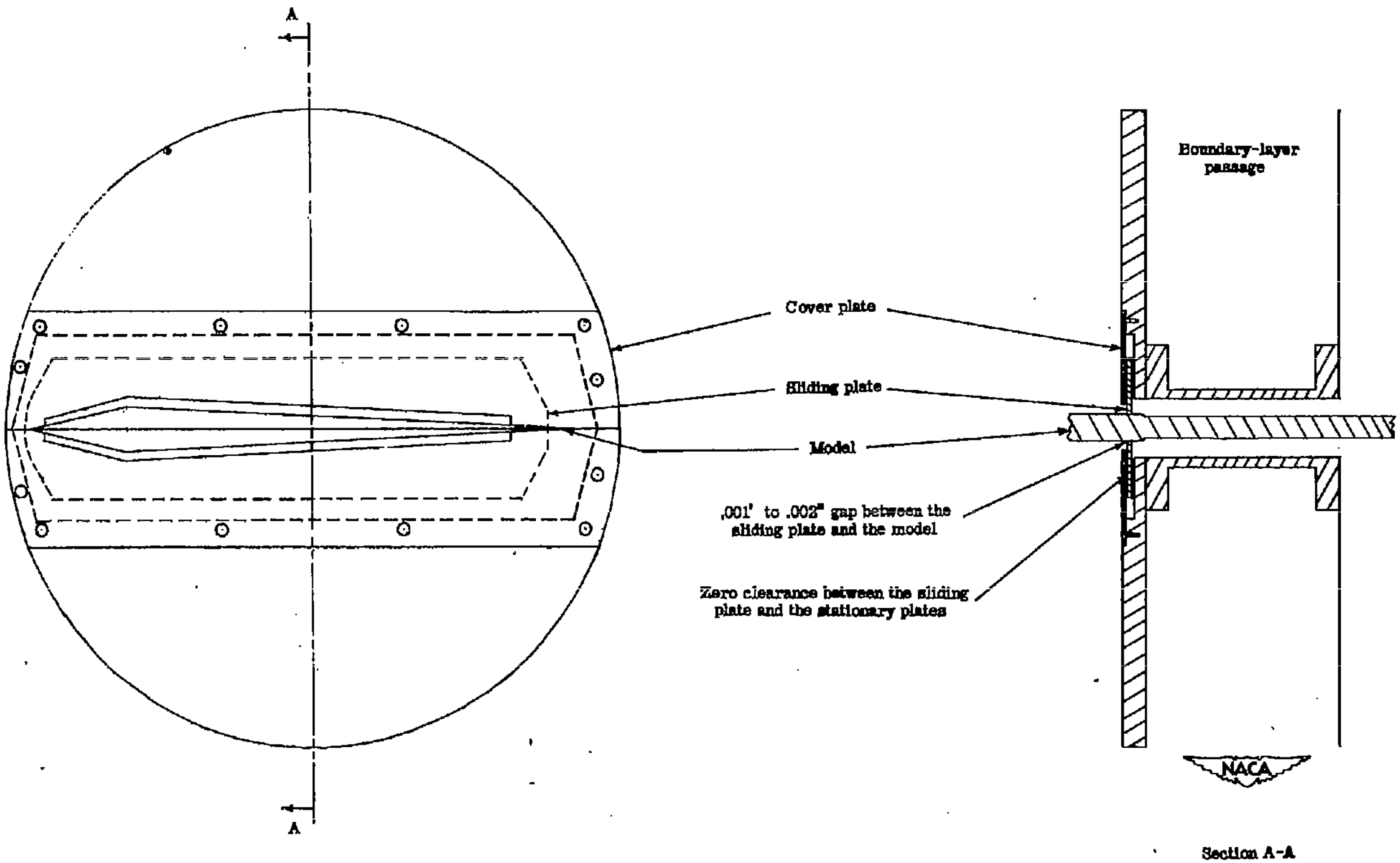
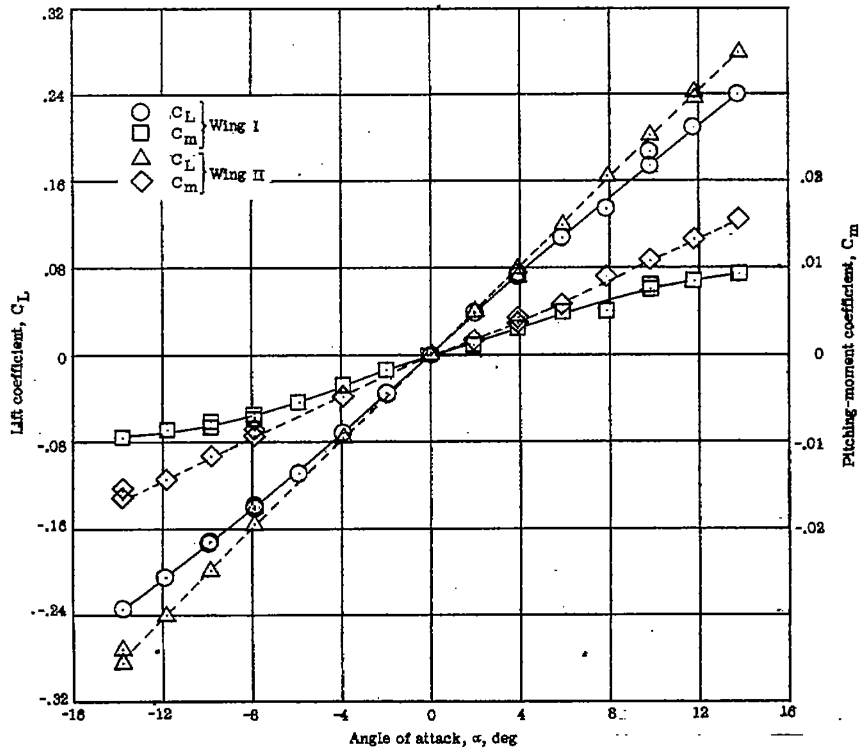
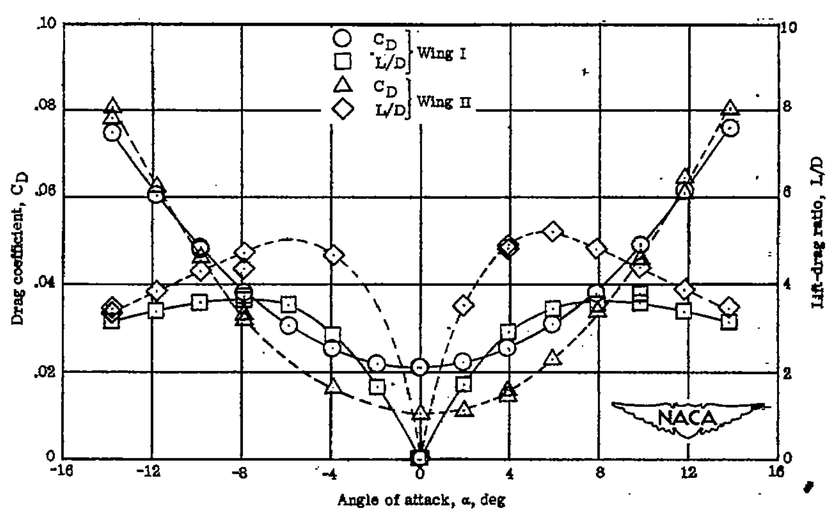


Figure 3.- Sliding-plate gap-sealing mechanism.

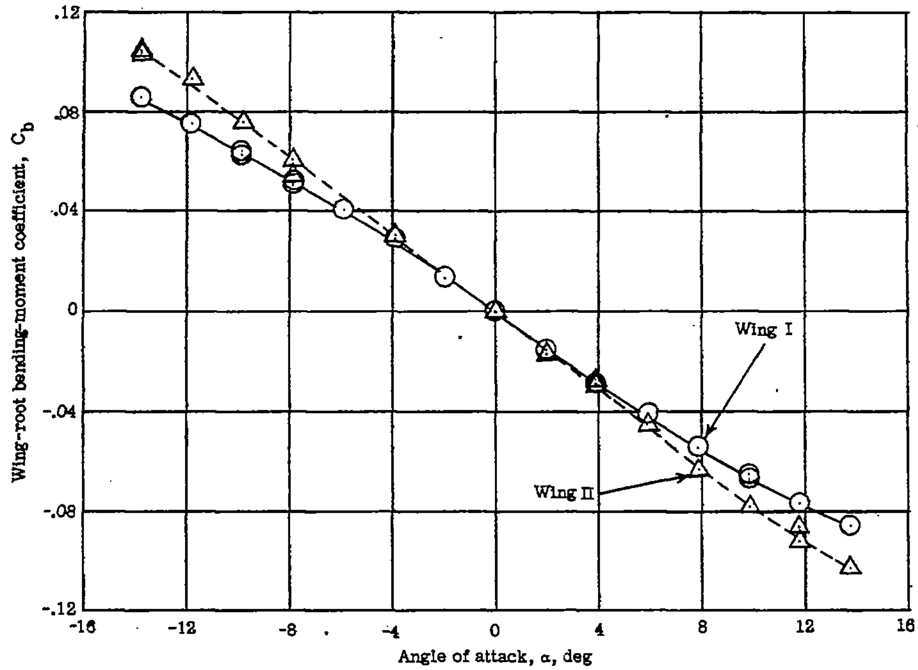


(a) Lift coefficients and pitching-moment coefficients.

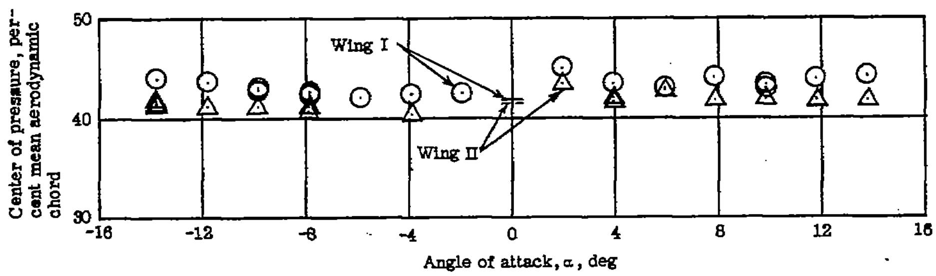


(b) Drag coefficients and lift-drag ratios.

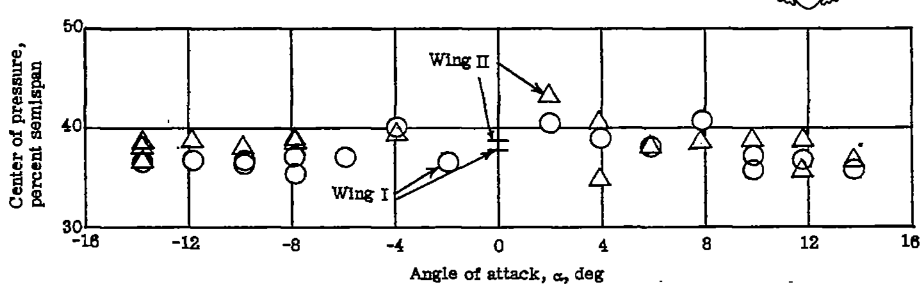
Figure 4.- Aerodynamic coefficients for 8-percent-thick delta wings at Mach number of 4.04 and Reynolds number 6.01×10^6 based on mean aerodynamic chord.



(c) Wing-root bending-moment coefficient.



(d) Chordwise center of pressure.



(e) Spanwise center of pressure.

Figure 4.- Concluded.

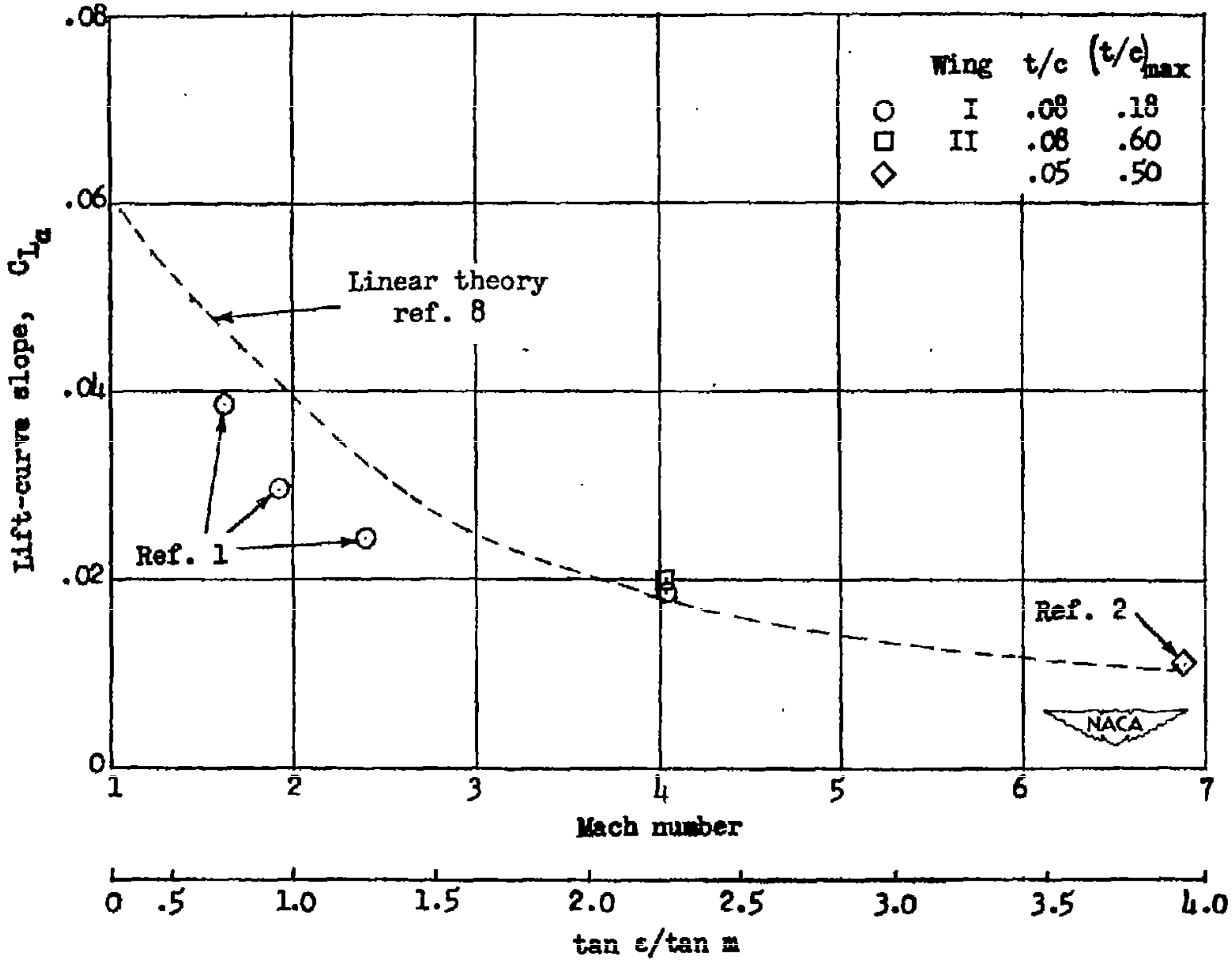


Figure 5.- Lift-curve slopes at zero angle of attack for aspect-ratio-2.31 delta wings with double-wedge airfoil sections at Mach numbers from 1.62 to 6.9.

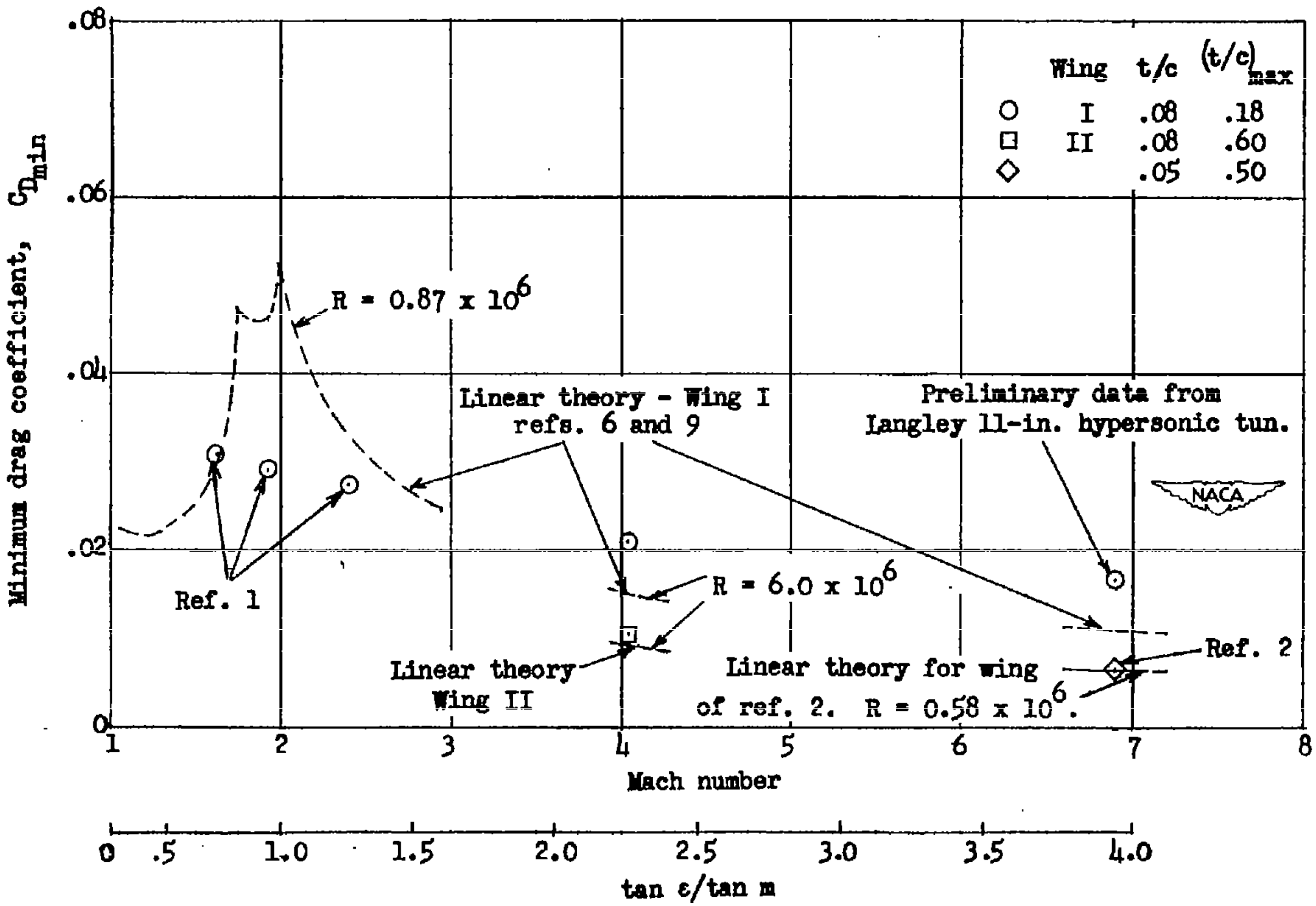


Figure 6.- Drag coefficients at zero angle of attack for aspect-ratio-2.31 delta wings with double-wedge airfoil sections at Mach numbers from 1.62 to 6.9.

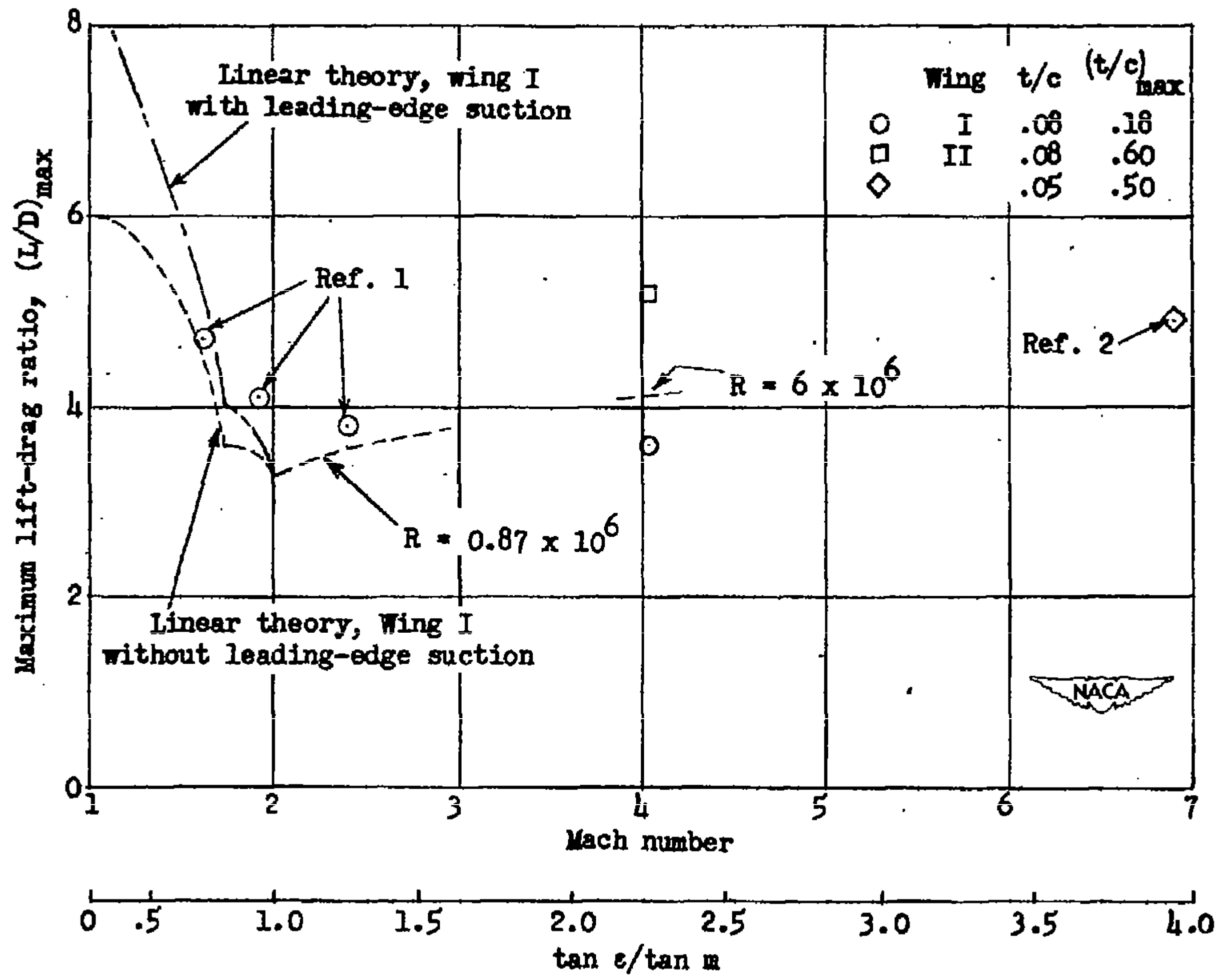


Figure 7.- Maximum lift-drag ratios for aspect-ratio-2.31 delta wings with double-wedge airfoil sections at Mach numbers from 1.62 to 6.9.

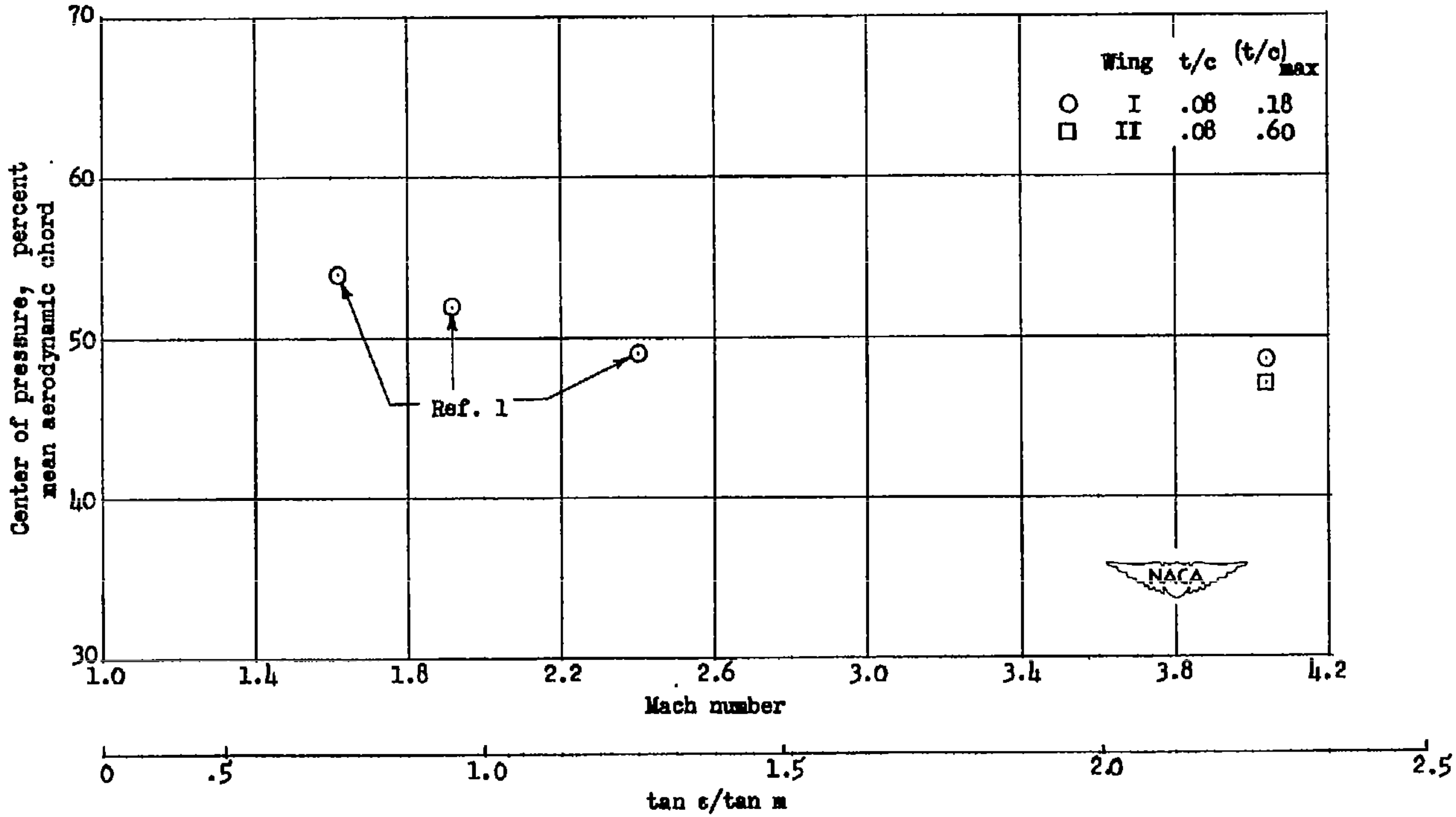


Figure 8.- Chordwise center-of-pressure locations at zero angle of attack for aspect-ratio-2.31 delta wings with double-wedge airfoil sections at Mach numbers from 1.62 to 4.04.

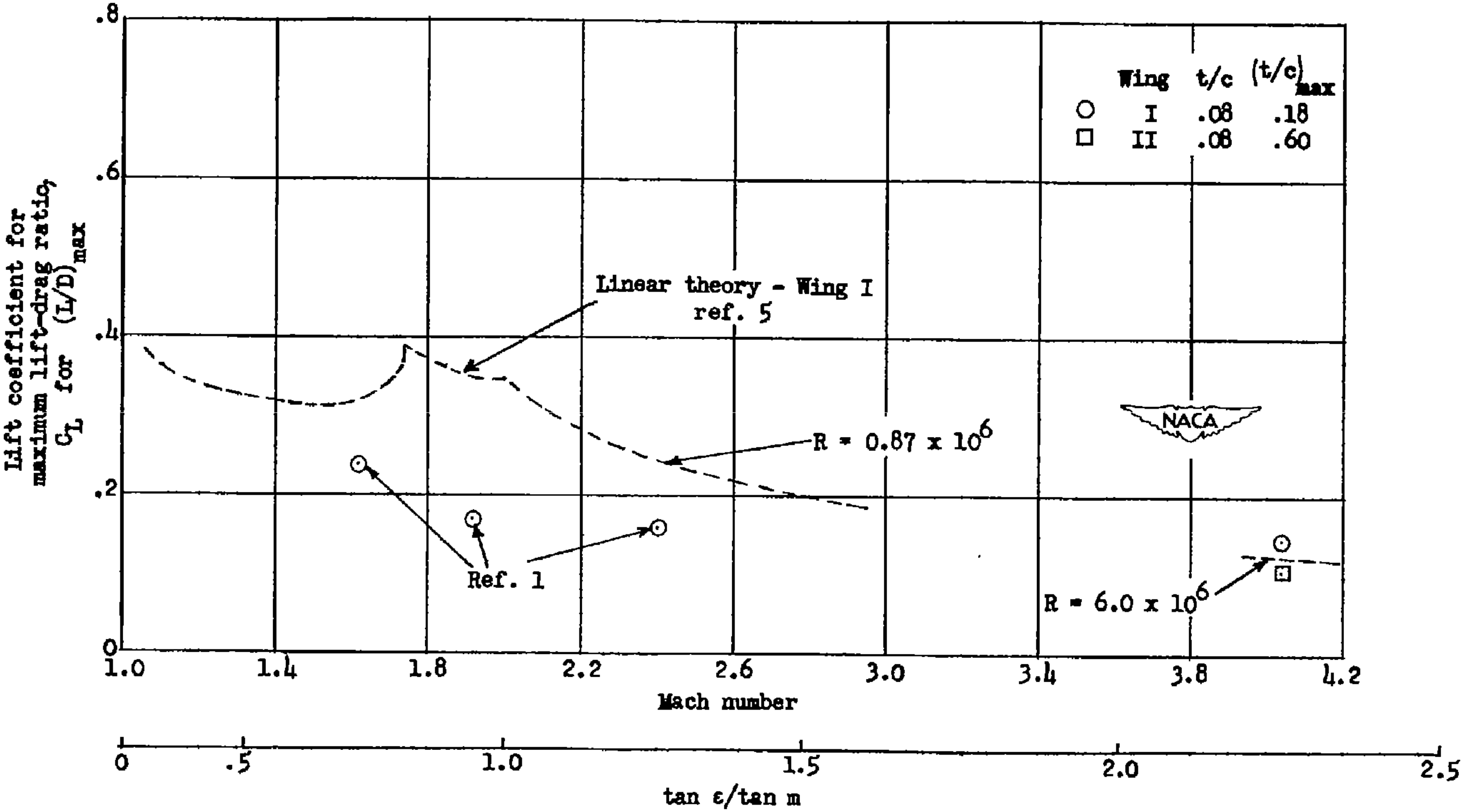


Figure 9.- Lift coefficients for maximum lift-drag ratios for aspect-ratio-2.31 delta wings with double-wedge airfoil sections at Mach numbers from 1.62 to 4.04.

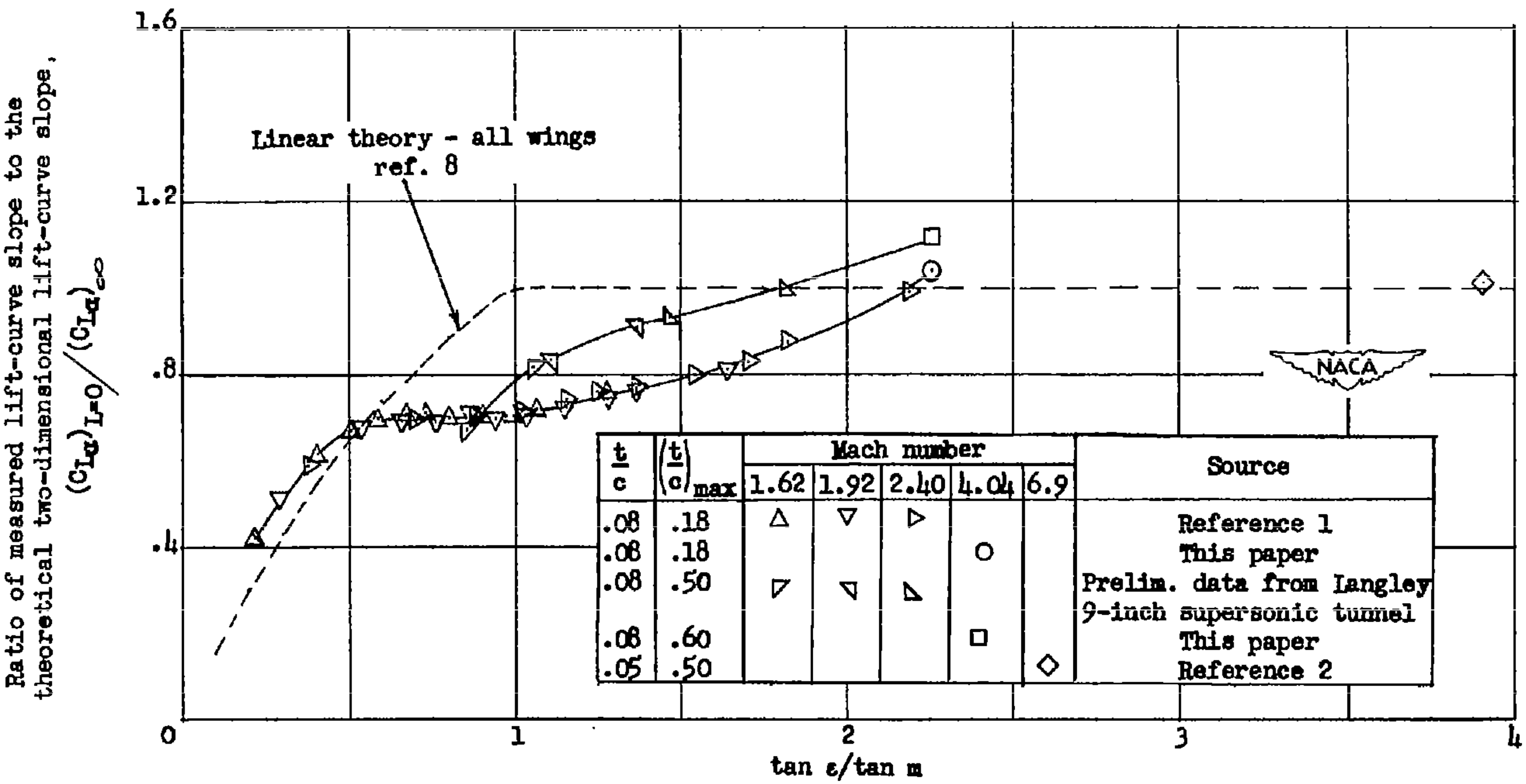


Figure 10.- Ratio of the measured lift-curve slopes to the theoretical two-dimensional lift-curve slopes at zero angle of attack for delta wings with double-wedge airfoil sections.

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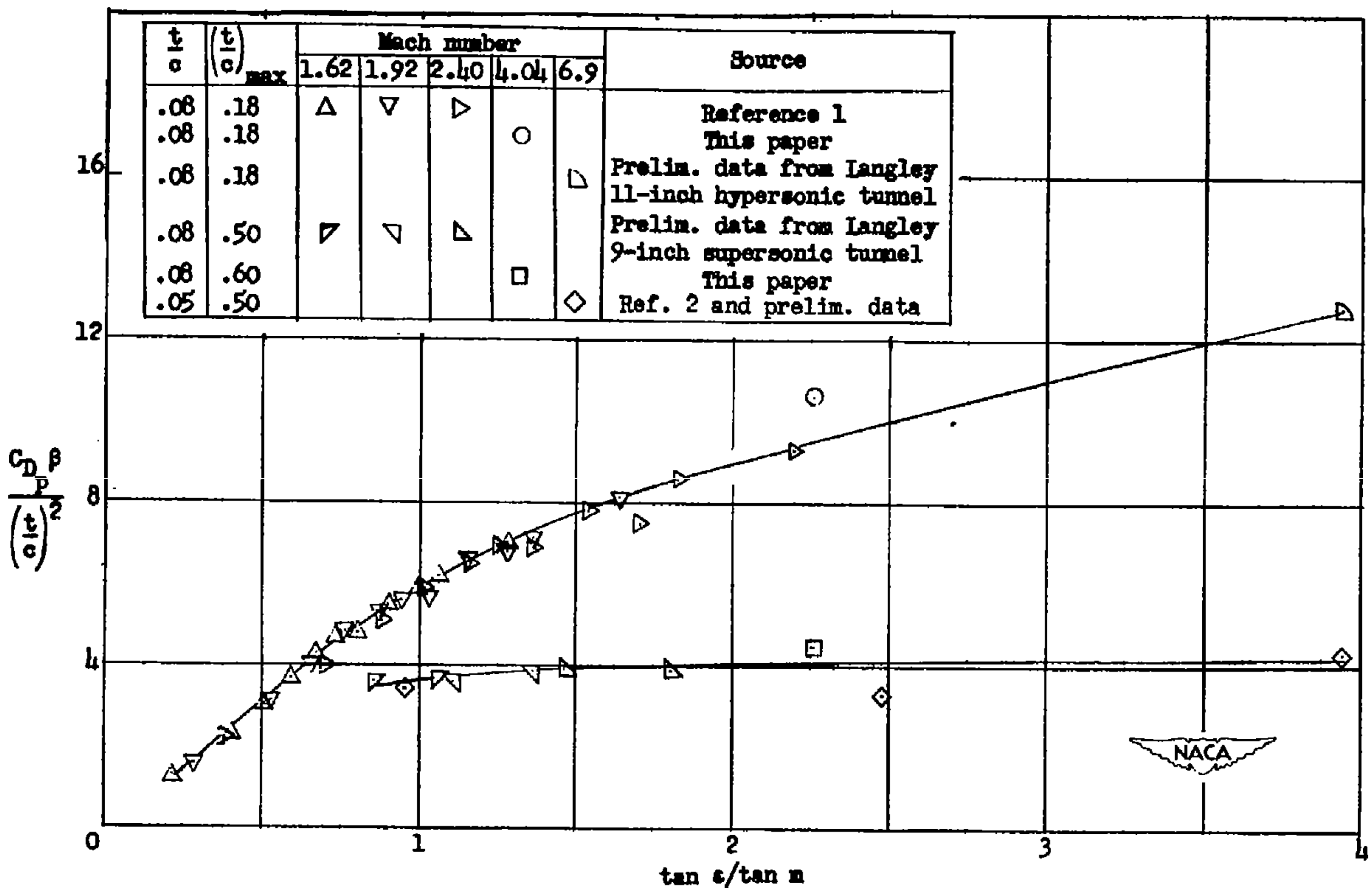

Figure 11.- Correlation of the experimental values of $\frac{C_{D_p}\beta}{(t/c)^2}$ for delta wings with double-wedge airfoil sections at Mach numbers from 1.62 to 6.9.